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September 1976

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ARMY MATERIALS AND MECHANICS RESEARCH CENTER
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Mathematical models
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Structural mechanics
Structural response

PREFACE

This document contains abstracts of research which were acceptable for presentation within the Work-In-Progress Sessions at the Army Symposium on Solid Mechanics, 1976. Unfortunately, not all of the papers could be scheduled for oral presentation at the symposium due to time limitations. Those which were not presented orally are so identified on page vi of the table of contents. These sessions were comprised of a series of brief presentations and discussions of current (but not necessarily complete) research relating to the theme of the conference: "Composite Materials: The Influence of Mechanics of Failure on Design." This meeting was held at Bass River (Cape Cod), Massachusetts on 14-16 September 1976. The proceedings of this symposium are published in a companion document: Army Materials and Mechanics Research Center Monograph Series Report, AMMRC MS 76-2, dated September 1976.

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AMMRC MS 74-9, September 1974, AD 786 524

Stress Analysis of Structural Joints and Interfaces -
A Selective Annotated Bibliography
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ARRANGEMENTS

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LARGE COMPOSITES STRUCTURES FOR THE BATTLEFIELD*

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ABSTRACT

Increased mobility on todays battlefield can be enhanced by a new class of ultra-lightweight heavy duty structures constructed of advanced composite materials. Such structures technology is being applied at General Dynamics Convair Division to mobile towers typical of towers used for radio antennas, radar terminals, satellite communication terminals, surveillance posts, etc. Large towers can be produced at one-third to one-fourth the weight of conventional steel structures - thus allowing for helicopter airlift. The airlift capability will allow completely assembled towers or tower modules to be dropped into high threat forward areas where close-in secure communications or surveillance are most needed. The reduction in cost, personnel hazard, and time for transportation to site and erection, combined with factors such as increased service life due to corrosion resistance and logistics considerations such as eliminating the need for building roads, make advanced composite battlefield structures potentially very attractive.

Convair's objectives for 1976 are to design a large portable tower, and to build and test major members of the structure. Objectives for 1977 are to build and test a full scale tower. This program will provide a more specific approach to determining the advantages, a detailed definition of problem areas, and an evaluation of a large composite structure in actual service.

Damage tolerance is a major consideration in battlefield equipment. A composite layup was developed to maximize resistance to the damage threats indentified for this type of structure. Failure mechanisms of damage and undamaged channel sections (10 inches wide by 30 inches long) loaded in compression are being investigated and compared by existing analysis methods. To date, damage equivalent to that imparted by the full force of a man wielding an 8-pound sledge hammer have been shown to cause only minor reductions in member strength. Joint strength is a second major strength consideration. A test program is in progress to define the strength of bonded joints, bolted joints, and bonded plus bolted joints. The third major failure mechanism of interest in designing a composite tower is overall stability. This is being studied with the NASTRAN computer program.

Structural members consist of box-I, wideflange-I, and hat sections. The composite layup is a hybrid of graphite, Kevlar, and epoxy. Metal fittings are used to join most composite members. The composite tower weighs only 15,000 pounds compared to 55,000 pounds for the equivalent steel structure--a weight saving of 73 percent.

* Work was performed under IRAD number 111-4760-760.

FACTORS AFFECTING THE STRENGTH OF ALUMINUM I-BEAMS SELECTIVELY REINFORCED WITH GRAPHITE/ALUMINUM

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ABSTRACT

The low density and high specific strength and modulus of graphite fiber reinforced aluminum make this a very promising material for selectively reinforcing structural aluminum alloys such as I-beams. This concept of selective reinforcement is particularly attractive because the reinforcement can be placed exactly where it is required thereby minimizing the cost and amount of reinforcement needed.

In order to demonstrate this concept, Fiber Materials, Inc., under an AMMRC contract has reinforced the flanges of four standard 6061 aluminum I-beams by hot isostatically bonding 0.040-in thick strips of Graphite/Aluminum composite to them. Theoretical calculations indicate that these reinforcing strips would increase the load carrying capability (bending moment) by 337%. These reinforced beams have been delivered to AMMRC where they are currently being evaluated. This evaluation program includes non-destructive testing followed by destructive testing in four-point bending to determine flexural strength and modulus, and flexural fatigue.

The non-destructive testing was conducted with X-rays and ultrasonic waves. The results of both tests correlated well and indicated that althouth the reinforcing layers showed regions which were well consolidated and fully bonded to the aluminum I-beam flanges, there were also areas with significant debonding. The latter condition will prevent the attainment of the theoretical strength of these beams.

The purpose of this presentation is to discuss these results and to indicate the factors which must be controlled in order to achieve full theoretical strengthening.

TRANSVERSE SHEAR STRESS FAILURE OF KEVLAR 49 COMPOSITE AND ITS EFFECT ON LIFE OF OVERWRAPPED PRESSURE VESSELS

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ABSTRACT

Pressure vessels overwrapped with filaments of Kevlar 49 have found increasing use because of their inherently high structural efficiency. For instance, the Space Shuttle Orbiter Vehicle utilizes this type of vessel to store high pressure gases such as helium and oxygen. The properties of Kevlar 49 make it possible to place the metal liner in a state of compressive stress as a result of a pressure sizing process. This permits the liner to operate with a lower tensile stress, thus enhancing its cyclic fatigue life.

Development tests and analysis have shown that a failure mode exists from the presence of high transverse-transverse shear stresses during the initial pressure sizing process. Since this sizing operation is mandatory to achieve vessel liner compressive preloading, the mechanics of this failure mode is of fundamental significance to the design of metal-lined composite vessels. These shear failures in the overwrap composite material cause the metal liner to work at a higher stress range thus reducing the fatigue life of the vessel.

Test are being conducted at the Johnson Space Center, Houston, Texas, on a representative filament wound metal-lined vessel to assess the effect of a low transverse-transverse shear ultimate strength on the design of these vessels. Additional tests on composite coupons are also being conducted to study the time related strains associated with the shear stress failure mechanism. Details concerning these preliminary tests will be presented and a stress analysis demonstrating the transverse shear stress failure and effects will be discussed.

VARIATIONS IN THE FAILURE MODE OF A 3-D CARBON-CARBON UNDER UNIAXIAL TENSILE LOADS

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ABSTRACT

The purpose of this research is to define the failure modes and failure envelope for a 3D carbon/carbon composite under uniaxial tensile loads. In particular, this study is investigating the response of the material at or near a free surface.

To this point, it has been established that two mechanisms or failure modes must be used to describe the failure envelope. Near a free surface, the mode of failure is shear. In this region the envelope can be described using a relationship of the form:

$$(UTS)_A = USS\ \phi\ X$$

Where $(UTS)_A$ is an apparent tensile strength, USS is the shear strength, ϕ is a material and geometry dependent term, and X is the distance to a free surface. Depending on the parameter ϕ, the shear effected region (defined by X) can be up to 2.0 inches.

At longer distances from the free surface, tensile failures become the predominate mode. Even in this region, the apparent tensile strength is still a function of distance from the free surface. In this region, the failure envelope can be described using a relationship of the form:

$$(UTS)_A = (UTS)_o\ 1 - (\psi e^{-\beta x})$$

where $(UTS)_o$ is an inherent tensile strength and the parameters ψ and β appear to be both material and geometry dependent variables.

The form of this second relationship could imply that even under "uniaxial" tensile loads, the 3D carbon/carbon does not exhibit plane strain due to shear lag. This in turn would imply that the entire tensile failure envelope is controlled by the shear strength and shear stiffness of the matrix. The study also indicates that the effective tensile stress-strain curve for the composite is also a function of the distance from a free surface.

These results could have a significant impact on the failure criteria used in finite element codes to predict the performance of 3D carbon/carbon composites in nosetip and rocket nozzle applications.

FAILURE CRITERIA FOR COMPOSITES UNDER IN-PLANE DISPLACEMENTS

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Y. Tirosh
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ABSTRACT

A unique computer-controlled in-plane loader has been developed with three independent actuators that apply tension, shear and rotational displacements to a specimen simultaneously and independently of each other. It is linked to a computer for experiment control, as well as data acquisition, storage and manipulation.

Tests are being conducted on a wide range of composite materials in order to develop data that will be useful in predicting failure behavior in structures. Initial studies are being performed on a graphite/epoxy cross-ply laminate in which the fiber orientation is changed in a regular fashion. The test data demonstrates good reproducibility. In addition, there appears to be a high degree of regularity in the data in that it changes in a smooth fashion as the displacements are changed continuously in a regular manner, and a distinct but continuous difference is observed for the different orientations tested. A finite element stress analysis routine is used to calculate the deformation field surrounding the notch tip in the test specimens, so that the local stresses at which fracture occurs could be calculated for each combination of loads.

The results will be extremely useful in the design of structural components. Future plans include demonstration of the validity of the fracture functions developed in laboratory tests on sub-components, and extension of the loading system to include out-of-plane loading.

BEHAVIOR OF ANGLE-PLY COMPOSITE LAMINATES
UNDER COMPRESSION IN THE THICKNESS DIRECTION*

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ABSTRACT

This work is part of a more comprehensive program of research into the thickness properties of laminated fiber composites. The angle-ply laminates approximate the material of filament wound tubes, a stress analysis of which requires knowdege of thickness properties. Thick angle-ply Kevlar 49/Epoxy laminates have been prepared and tested to failure in compression along the thickness direction. The ply angles were 0° (undirectional), ± 15°, ± 30°, and ± 45° (cross-ply). The stress-strain behavior as well as the failure modes showed striking differences. The 0° laminate exhibited linear stress-strain response and failed at a low fracture stress. The fracture was by shear through the resin matrix. The stress-strain behavior of the ± 15° material was linear at first followed by considerable flattening of the curve during a stage of extension plastic deformation and finally an upturn of the curve ending with fracture at a moderate stress value. This sample experienced expansion and barrelling in the 90° direction, with contraction and necking in the 0° direction. The fracture was by shear along a plane making approximately 30° with the compression axis and containing the 0° axis. Some delamination also took place. Furthermore, the ply angle increased from 15° to around 26° on the surface layers and approached 45° in the middle of the midplane. The fracture obviously involved both matrix and fiber breakage. The ± 30° laminates showed similar but less marked effects. The failure which occured at a higher stress was by a combination delamination and shear. The cross-ply laminates exhibited practically linear stress-strain response to failure. The cross section dimensions remained unchanged and failure at a much higher stress occured by delamination with plies being ejected out of the laminate. Optical and SEM micrographs are presented and the behavior discussed in terms of the interaction between constituents and plies.

*This work was supported by AMMRC and ARO

MODELING OF DAMAGE AND FAILURE IN NOTCHED COMPOSITE LAMINATES

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ABSTRACT

Research is currently being conducted on the influence of various manufacturing and service flaws on the strength and durability of advanced composite laminates. Although in many cases of through-the-thickness flaws the "average stress criterion" has been shown to be effective in predicting failure,* this criterion does not take into account the progressive failure process that is known to occur in composites. It is expected, therefore, that there are cases such as general biaxial loading of a flaw where the direction of flaw growth is not known apriori that will resist prediction by this method. To overcome this difficulty, as well as to lend insight into the failure process, a new method is currently being studied which is capable of modeling and tracing the damage accumulation in notched or unnotched composite specimens that ultimately leads to failure. The idea is, quite simply, to model the actual constitutive behavior of the laminate both before and after damage has occurred. This allows the ultimate failure of notched laminates to be predicted under general in-plane loading based only on a knowledge of the individual ply behavior.

The development of the incremental finite element program necessary to perform the calculations has been finished and, to date, two examples have been run. These involve failure predictions for two different graphite/epoxy laminates containing holes and subjected to uniaxial tensile loads. The strength predictions made by computer agreed very well with the test results. Currently, more examples are being run and refinements are being made to the program itself.

---

* R. J. Nuismer and J. M. Whitney, "Uniaxial Failure of Composite Laminates Containing Stress Concentrations", Fracture Mechanics of Composites, ASTM STP 593, 1975, pp 117-142.

POST-BUCKLING FATIGUE BEHAVIOR OF ADVANCED COMPOSITE SHEAR PANELS*

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ABSTRACT

The objective of this ongoing analysis and test program is to evaluate the post-buckling behavior of graphite/epoxy shear panels subjected to static and fatigue loads. The analytical evaluation of post-buckling strength, involving nonlinear deformation analysis, is performed with use of the STAGS (Stress Analysis of General Shells) finite-difference computer program. The experimental evaluation of post-buckling behavior of unstiffened flat laminated shear panels under static and fatigue loads has been completed. The post-buckling evaluation of stiffened panels is in progress. The influence of laminate thickness and material properties on post-buckling behavior has also been considered. The unstiffened 6-inch by 18-inch laminated graphite/epoxy shear panels were tested under static and cyclic load in the post-buckling range. The two materials considered were unidirectional 3501-5/AS graphite/epoxy and woven HMF330C/34 graphite epoxy. To evaluate the influence of various geometric parameters, a total of five different material-laminate combinations, consisting of three unidirectional graphite/epoxy laminates of 4-ply $(\pm 45)_s$, 8-ply $(\pm 45,0,90)_s$, and 12-ply $(\pm 45_2,0,90)_s$ and two woven graphite/epoxy laminates of 2-ply $(45)_s$ and 4-ply $(45,0)_s$ were evaluated. Static failure loads ranged from five to twenty times the initial buckling loads depending on the panel thickness. Static strength values ranged from 39 percent to 67 percent of the unbuckled laminate strength. Identical panels were subjected to 10^6 cycles of constant amplitude fatigue loading with loading ratio of 0.1. The maximum load was set at 67 percent of the panel static strength. After fatigue exposure, panels were static tested to failure.

The significant conclusions and design recommendations for unstiffened panels are summarized below:

The flat laminated composite panels have significant postbuckling strength capability both under static and fatigue loads. Post-buckling strength values depend on the panel geometry and up to 20 times the buckling load have been achieved.

The buckled panel strength is always less than the unbuckled basic laminate strength. The loss of strength is a function of the panel geometry, laminate thickness and ply orientations. Post-buckling strength values of 39 percent to 67 percent have been observed in the tests.

Fatigue cycling in the post-buckling load range, applying 10^6 cycles at 67 percent of the panel static strength, does not degrade the composite laminate residual strength capability.

For prediction of post-buckling behavior, numerical analysis methods such as the STAGS computer program have to be used. The numerical approach provides a significant capability for this complex problem. However, the numerical approach requires large computing times.

*This work is being done as part of Northrop Corporation, Aircraft Division's Independent Research and Development program.

ANOMALOUS FATIGUE BEHAVIOR OF GRAPHITE-EPOXY COMPOSITES

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ABSTRACT

Flat, doubly-notched specimens of 0°, ±60° graphite-epoxy composite specimens were subjected to high stress level, zero-to-tension cyclic loading. During the tests periodic photomicrographs of both notch roots were taken using a dual automatic photomicrographic system. The system includes reflection microscopes, electronic flash light sources, electrically operated 35mm cameras and programming components to allow exposures to be made at maxium tension load at predetermined cyclic times without interrupting the fatigue test. Prior to testing, the notch roots of the specimens were polished in a polishing fixture using aluminum oxide polishing compound. As a result of the polishing process, the individual 8 micrometer diameter graphite fibers were clearly visible and matrix voids and fiber debonding could easily be seen.

Anomalous fatigue behavior was observed in the limited number of tests which were performed. In two tests where the cyclic loads were approximately 75% and 90% respectively of the static fracture load there were differences in the crack initiation behavior. In the specimen which was tested at 75% of the static load, a macrocrack developed in one of the 60° layers during the first cycle of loading yet the specimen was then able to sustain 10,000 cycles without fracture. In the specimen which was tested at 90% of the static load the observations indicated progressive damage during the test which included fiber cracking in the 0° layers, fiber debonding in the 60° layers, and delamination between a pair of adjacent 60° layers. This specimen also sustained 10,000 cycles of cyclic loading. Both specimens were then monotonically loaded to failure and their residual strengths were approximately the same as that of a virgin static specimen.

A third specimen, which we had planned to cycle at 90% of the static fracture load failed on the first cycle of the test. A macrocrack developed at 55% of the static fracture load and catastrophic failure occured at 78%.

Possible causes for the apparent anomalous behavior have not yet been isolated however there are indications that manufacturing irregularities in the material may contribute to the strange fatigue behavior.

IMPROVEMENT OF FATIGUE LIFE OF BORON-EPOXY LAMINATES BY PROLONGED HEAT UNDER LOAD

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ABSTRACT

One mode of fatigue damage in laminated composites is believed to originate from matrix degradation. The rate of degradation is sensitive to the amount of residual stresses left after the curing cycles; the fibers are left in compression and the matrix in tension. It is conceivable that if the residual stresses are eliminated the fatigue life of laminated composites can be lengthened.

In the present work, the investigators attempted to alleviate the residual tension stress in the matrix by prolonged loading of the laminates at elevated temperatures. Boron-epoxy laminated composite specimens with a circular hole were used for this investigation.

During the fatigue tests, thermograms mapped out the temperature distribution which, in turn, indicated the extent of damage in the laminate matrix. Untreated specimens exhibited a very early and subsequently very rapid temperature rise. In contrast, the treated specimens stayed cool for a substantially longer period and the temperature rise was much slower.

On the average, the heat treatment lengthened the fatigue life by an order of magnitude. The effect of such treatment on stiffness, fracture-strength and the fatigue degradation rate in terms of broken fibers will also be reported.

EFFECT OF VOID CONTENT ON THE BEHAVIOR OF ADVANCED FIBER REINFORCED COMPOSITE MATERIALS

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ABSTRACT

Current Air Force structrual integrity policy requires that all "safety of flight" structure on new Air Force aircraft will be designed to be damage tolerant, that is, it requires that the structure shall have a specified minimum residual strength (with damage present) both during and at the end of a specified period of unrepaired service usage. Compliance with this policy requires: (a) the ability to predict failure modes, (b) the ability to predict residual strength levels in the presence of probable types of material or manufacturing deficiencies and in-service damage, and (c) the ability to predict degradation in residual strength as a function of "real time" operational usage.

One prevalent manufacturing defect is high void content in the whole or a small portion of a structural laminate. High void content has a number of causes, for example, loss of or nonumiform pressure during the cure cycle and use of over aged prepreg tape. While having only a slight effect on "fiber dominated" properties, high void content can have a tremendous effect on "matrix dominated" properties. For example, the interlaminar shear strength is decreased by about 40% by a void content of about 10%.

Most of the available investigations that address the effect of void content on the behavior of advanced composite laminates were performed as a part of manufacturing methods development programs and, hence, are buried in reports not dealing specifically with the effect of void content. In the present work, information on the effect of void content taken from the hardware and manufacturing methods reports as well as the specific programs addressing the void content question will be discussed in detail. The paper will specifically address the following questions:

(a) What are the causes of high void content in laminates?
(b) What quality assurance procedures are available to guarantee laminates with low void content?
(c) What nondestructive inspection methods are available to check the void content of laminates? and
(d) What is the effect of voids on both the static and long time properties of laminates?

DAMAGE TOLERANCE OF COMPOSITE SANDWICH STRUCTURES SUBJECTED TO BALLISTIC AND LASER IMPACT

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ABSTRACT

An investigation is underway to determine the damage tolerance of composite sandwich structures for 23mm HEI ballistic and laser damage. Sandwich panels, box structures, and cylinders are being tested. Panel dimensions are 40 x 22-in. All panels have [±45°] laminated face sheets and 1.0-in deep core. Face sheet materials for the panels are SP250-SF1 (Scotchply) fiberglass or 5208 MOD I graphite/epoxy. Core materials used are Nomex honeycomb, and polyurethane or polymethocrylinide foam. Box structures of various sizes were built up from panels and were tested to determine the difference in damage sustained due to blast confinement as compared to unconfined blasts on flat panels. The composite cylinders tested to date had fiberglass inner and outer face sheets and Nomex honeycomb core. These were 24-in in diameter and 42-in long.

Panels have been impacted while under static tension load of approximately 30% of undamaged ultimate strength, and the residual strength of impact damaged panels is determined after testing. In the ballistic tests the point of detonation of the impacting round is varied in order to achieve different combinations of blast and fragment damage. Laser tests are conducted with varying impulse duration, size of impact area, and pulse intensity. Test results to date show that the degree of damage due to 23mm impact varies significantly with location of the blast and was generally more severe than would be expected considering the loss of structural material alone. Panels with PMI foam cores sustained less blast damage (i.e., face sheet debonding) than panels with Nomex cores in the ballistic tests and also appeared to more damage tolerant to laser irradiation. Panels in box sturctures suffered greater blast damage than flat panels, while the cylindrical structures appeared to have the greatest damage tolerance of any of the configurations tested. The composite structures, in general, exhibited as good or better damage tolerance than equivalent sandwich structures with metallic face sheets.

FAILURE ANALYSIS OF COMPOSITE CYLINDERS SUBJECT TO INTERNAL BLAST

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ABSTRACT

The REPSIL shell code is being used to analyze the blast response of advanced composite and aluminum cylinders of the same length, radius, and weight. REPSIL is a finite difference computer program which calculates the large deflection transient motion of thin Kirchhoff shells. The program marches out the solution timewise by cyclically solving an explicit, time centered difference formula for displacements. REPSIL was modified to permit the analysis of shells with orthotropic skin properties.

The first configuration studied was a point internal blast detonation located at the geometric center of a thin monocoque cylinder. The variables were the skin material and thickness and the mass of the charge blast. The composite skins were composed of uniaxial plies stacked in the axial (0°), diagonal (±45°), and hoop (90°) directions. The table below shows the ratio of the critical charge mass M needed to fail high strength graphite (HSG)/epoxy or boron/epoxy cylinders to the critical charge mass M_{al} needed to fail an aluminum 6061-T6 cylinder. The cylinders studied all failed from excessive tensile stress in the hoop direction. The table shows the increase in M/M_{al} for HSG/epoxy shells when the percentage of hoop plies is increased from 25 to 55%. Further increases in the percentage of hoop plies would reduce the bending stiffness EI and torsional stiffness GJ of the HSG/epoxy shell below the corresponding stiffnesses (EI_{al}, GJ_{al}) of the aluminum shell. For the configuration studied, the critical charge masses for HSG/epoxy and boron/epoxy cylinders are less than half the critical charge mass for an Al 6061-T6 cylinder of the same weight, bending stiffness and torsional stiffness.

Design recommendations for blast tolerant composite helicopter tail cones and plans for more detailed analytic and experimental study will be discussed.

	HSG/Epoxy		Boron/Epoxy
% 0°, +45°, -45°, 90° plies, respectively	25, 25, 25, 25	15, 15, 15, 55	11, 11, 11, 67
M/M_{al}	.33	.41	.36
EI/EI_{al}	1.4	1	1

FRACTURE CONTROL FOR COMPOSITE LAMINATES

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ABSTRACT

The damage-tolerance criterion used in metallic aircraft structure requires that a damaged structure sustains design limit load following the loss of a structural element due to a fatigue failure. This can be achieved by reducing the stress level and hence the stress-intensity factor at the crack tip by providing alternate load paths. The use of stiffeners in panels and crack stopper straps in panelized wing and fuselage are the examples. The need for effective fracture control for composite structures became even more clearly drawn when it was learned that advanced composite materials show little or no observable crack growth prior to rapid fracture under cyclic loading. This brittle nature of failure exhibited by the composite materials requires that some degree of fail-safety again be employed to prevent catastropic failure due to overload or damage incurred during service.

The Lockheed-Georgia Company initiated a study in 1975 under an in-house funded IR&D program to assess the effectiveness of buffer (softening) strips in structural components as crack-arrest mechanisms in fail-safe design of composite structures.

Based upon the tests of 8-ply $[\pm 45, 90, 0]_S$ graphite/epoxy laminates conducted at Northrop under Air Force Contract F33615-74-C-5182, it was concluded that the E-glass/epoxy material provided the best crack arrestment capability. However, the results obtained at Lockheed-Georgia for 16-ply $[\pm 45, 0_2]_{2S}$ graphite/epoxy laminates indicated that both Kevlar and E-glass have good capability for arresting the rapidly propagating crack, and Kevlar offers better arrest potential than E-glass.

Under a current program, wider specimens containing four buffer strips are being fabricated and both static and fatigue tests will be conducted to determine if there is a second crack arrest. Efforts to develop a design concept based on the classical laminate theory and linear-elastic fracture mechanics are continuing.

RESEARCH ON THE DAMAGE TO AIRCRAFT COMPOSITE MATERIAL FUEL TANKS DUE TO PENETRATING PROJECTILES (HYDRAULIC RAM) AT THE NAVAL POSTGRADUATE SCHOOL

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ABSTRACT

Metal aircraft fuel tanks that are subjected to ballistic impact and penetration by small arms fire and missile fragments can be severly damaged, with large petalling of the tank walls occurring at the entrance and exit points of the projectile. The damage mechanism, called hydraulic ram, is a very high pressure wave in the fuel caused by the passage of a ballistic penetrator through the fuel. Fluid pressures and wall strains have been experimentally measured and analytically predicted at NPS for rectangular tanks with aluminum walls. A summary of this work is given in Ref. 1. Due to the fact that aircraft fuel tanks made of composite materials are now being seriously considered, the effect of hydraulic ram on composite material tank walls has been investigated. In Ref. 2, the various effects of hydraulic ram on a clamped 11-in. square, 0.067-in. thick, graphite/epoxy wall due to penetration by a .22 caliber projectile were examined. Shots at 2600 fps caused only light damage to the plate. At 2800 fps, the hydraulic ram caused considerable damage, including total severance of the plate from its clamped support over much of the outer perimeter.

The objectives of the research in Ref. 3 were to show the relative importance of the transverse shearing forces produced by hydraulic ram loading on military aircraft fuel tank joint designs for composite materials, and to propose fuel tank test section designs based upon specific composite material fuel tank design concepts for the F-16, F-18, and a Navy V/STOL delta wing. With the use of a finite element analysis, the transverse shearing force at a metal fastener was shown to be a major cause of failure at the attachment, primarily by an out-of-plane push-out mode of failure. This type of failure could have a significant effect on the structural integrity of a major load carrying member of the aircraft, such as the wing box beam. In this situation, a large portion of the wing skin over the fuel tank may become detached from the spars, ribs and stringers, causing a serious degradation in the strength and stiffness of the wing. Future research will be devoted to the study of the amount of resistance to out-of-plane push-out shown by various composite attachment designs and to the determination of the amount of area over which push-out may occur. This work has been supported by the Joint Technical Coordinating Group for Aircraft Survivability and the Naval Weapons Center, China Lake, California.

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ON THE ANALYSIS OF NONHOMOGENEOUS CYLINDRICAL SHELLS

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ABSTRACT

In order to guide the design of composite material structures towards a more analytical basis, technical theories which have been checked experimentally and/or by comparison with three-dimensional solutions are needed. Much of the available work in this area, however, deals only with "thin" plate and shell theories. A summary of this literature can be found in an excellent recent article by Bert [1]. In the present paper both "thin" and "thick" shell theories for nonhomogeneous, anisotropic, elastic materials are derived and their validity and range of applicability tested by comparison to exact elasticity solutions.

The first part of this paper considers the problem of a multilayered composite cylinder under internal and external pressure, stretching and torsion. The asymptotic method [2] was employed to derive the appropriate first approximation (thin shell) and second approximation (thick shell) theories governing the asymmetric deformation of nonhomogeneous anisotropic cylinders. The advantage of this method of deriving shell equations is that it yields expressions for all the stress components directly, including the transverse ones. This is of importance when performing a failure analysis.

To analyze the effect of edge fixity and loading, the problem of a fixed-free composite cylinder with applied free end moment and shear force and a hydrostatic loading is analyzed. The asymptotic method is again employed to derive the relevant thin and thick shell equations. However, the physics of the problem dictates the use of length scales which are different than those employed in the first problem.

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SPECIAL FINITE ELEMENTS FOR THE ANALYSIS OF FLAW GROWTH AND FRACTURE IN FIBER COMPOSITES

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ABSTRACT

A research program involving the development of special finite element techniques for the study of crack problems in fiber composites is in its second year of sponsorship under a grant from the NASA-Lewis Research Center. The primary objective of the study is to extend present analytical capabilities for solving crack problems in composites by the development and refinement of a three-dimensional finite element analysis. The initial analysis, presently near completion, will treat surface, embedded, or through-thickness cracks, modelling each ply of the composite as a homogeneous, elastic continuum. The analysis is based on the hybrid stress model formulated through the Hellinger-Reissner Variational Principle, and employs eight-node hexadron elements. Stress distributions will be obtained for several ply configurations and crack geometries, and appropriate local failure criteria will be employed to predict damage extension and fracture. Future efforts will include the development of higher-order elements for the treatment of nonlinear effects. The research program also includes an experimental phase whose objective is to obtain damage extension and fracture data for several crack geometries and ply configurations to be used as test cases for the analysis.

A FUNDAMENTAL ANALYSIS FOR CRACK GROWTH IN A FIBER REINFORCED COMPOSITE

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ABSTRACT

This abstract describes work in progress to develop an analysis procedure for crack growth in a fiber reinforced composite material. The work is supported by NASA, Ames Research Laboratory.

Unlike metals, fracture in composite materials is determined by a sequence of distinct micromechanical processes. The long-range goal of this work is a mathematical model that will permit predictions of the strength of fiber reinforced composites to be made from the basic mechanical properties of the constituents. The approach merges a general anisotropic homogeneous material representation with a local heterogeneous region (LHR) surrounding the crack tip.

The unique feature of the mathematical model described in this paper stems from the interactive manner in which individual local rupture events are permitted to occur in the local heterogeneous region. Specifically, the model has the capability to represent the fracture process as a sequence of localized events determined from the analysis. The LHR consists of elements representing the matrix, the fibers, and the fiber-matrix interface. The basis for the prediction of local rupture is related to an intrinsic critical energy density. In the model, each constituent (and interface) of a fiber composite can rupture independently at any point where the energy criterion is met.

Previously, computations were performed for unidirectional composites under tensile loading using assumed properties for the composite constituents. It was shown that fiber breakage, crack bridging, matrix-fiber delamination, and axial splitting can all occur during a period of (gradually) increasing load prior to catastropic fracture. Of most importance, the computations revealed the sequential nature of the stable-crack growth process in fiber composites.

In this paper, comparisons of fracture stress values obtained in graphite/epoxy composites with the predictions of the model are made. The experimental results of Brinson and Yeow* on unidirectional composites using double edge notch tensile coupons were used. The calculations are performed using the results of finite-element calculations on double edge notch specimens with various crack length to specimen width ratios and crack angles. From these results, the displacement field to be imposed at the outer boundaries of the LHR is determined. Then, separate LHR calculations are performed for each experiment of interest. By forcing the computational result to match one experiment, the critical energy level for crack growth can be established. This value is then used to predict the failure loads in other cases.

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